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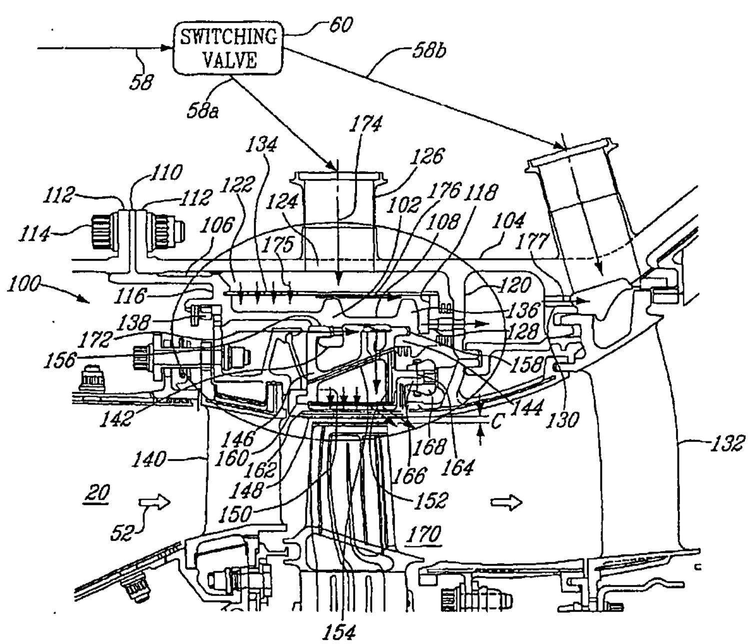
(54) Title: HYBRID TURBINE BLADE TIP CLEARANCE CONTROL SYSTEM



(57) **Abstract:** A turbine shroud cooling system used in a gas turbine engine for controlling tip clearance between a turbine shroud assembly and turbine rotor blades comprises a cooling air passage for selectively directing a cooling air flow between components to be cooled and a turbine shroud support assembly for controlling the tip clearance and then later re-directing the cooling air flow to cool a downstream turbine component.

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HYBRID TURBINE BLADE TIP CLEARANCE CONTROL SYSTEM

FIELD OF THE INVENTION

[0001] The present invention generally relates to gas turbine engines, and more particularly to clearance control between turbine rotor blade tips and a stator shroud assembly radially spaced apart therefrom.

BACKGROUND OF THE INVENTION

[0002] A gas turbine engine includes in serial flow communication, one or more compressors followed in turn by a combustor and high and low pressure turbines, disposed symmetrically about a longitudinal axis centerline within an annular outer casing.

[0003] Each of the turbines includes one or more stages of rotor blades extending radially outwardly from respective rotor disks, with the blade tips being disposed closely adjacent to a turbine shroud assembly supported within the casing. It is desirable to maintain the gap between the blade tips and the shroud assembly as small as possible throughout the engine operation range because the combustion gas flowing therethrough bypasses the turbine blades and therefore provides no useful contribution. However, because the material of the stator components and the turbine rotor are different, and because inertia has an influence on the expansion of the rotor, the stator components, i.e. the engine case, outer air seal, and support mechanism, expand at a different rate than the expansion of the rotor. Therefore, the gap must be sized larger than would otherwise be desirable.

[0004] Conventionally, small gas turbine engines typically use a passive tip clearance control system when attempts are

made to optimize the thermal response characteristics of the rotor and the casing. Full pressure compressor air is used both as the cooling medium and as the air seals around the blade tips, and is then exhausted into the turbine combustion gas path. When operating the engine during a transitional period, the thermal response rates of the casing and the rotor blades are difficult to match, thereby resulting in a pinch-point. This pinch-point causes a system limitation as to the minimum achievable tip clearance without rubbing.

[0005] Larger engines usually use active tip clearance control where inter-stage compressor bleed air is used to externally cool the turbine casing, typically in an impingement manner. This inter-stage compressor bleed air can be turned off by a valve during initial operation so as to avoid the pinch-point. When the engine has thermally stabilized, the valve is opened and the turbine casing effectively contracts to minimize tip clearance. Typically, this inter-stage compressor bleed air is dumped into the nacelle and lost to the cycle after having cooled the turbine casing.

[0006] Various efforts have been made to improve turbine tip clearance control in gas turbine engines. Examples of those efforts are illustrated in United States Patent 4,513,5697 to Deveau et al. on April 30, 1985; United States Patents 5,593,277, 5,562,408 and 5,553,999 issued to Proctor et al. on January 14, 1997, October 8, 1996 and September 10, 1996 respectively; United States Patent 5,048,288, issued to Bessette et al. on September 17, 1991; United States Patent 4,358,926, issued to Smith on November 16, 1982; and United States Patent 6,487,491, issued to Karpman et al. on November 26, 2002. These prior art patents disclose method, systems and apparatuses for improving turbine tip clearance control in one or more aspects of this matter. Nevertheless,

continuous efforts to develop the technology in this field are still needed in order to achieve better performance of gas turbine engines, particularly for use with aircraft. The prior art offers complex solutions and solutions which do not maximize the efficiency of cooling air systems in the engine. Improvements are therefore desired.

SUMMARY OF THE INVENTION

[0007] One object of the present invention is to provide a turbine tip clearance control system for improving tip clearance control preferably without extra cooling air consumption, thereby improving overall gas turbine engine performance. Other objects will be apparent from this disclosure.

[0008] In accordance with one aspect of the present invention, there is provided a turbine shroud support configuration used in a gas turbine engine, for supporting a turbine shroud assembly having a plurality of turbine shroud segments radially spaced apart from a plurality of turbine rotor blades, and further for controlling tip clearance therebetween. The turbine shroud support configuration comprises an annular housing adapted to be secured within a turbine support case, and means attached to an inner side of the annular housing for supporting the respective turbine shroud segments in place. A first cooling air passage is provided for directing a first cooling air flow passing therethrough, thereby controlling the tip clearance between the turbine shroud segments and the turbine rotor blades. The first cooling air passage is defined at least by the annular housing and is isolated from a combustion gas path defined within the turbine shroud assembly. The first cooling air passage is in fluid communication with a downstream cooling air passage of the gas turbine engine for

further directing the first cooling air flow to cool a turbine component downstream of the turbine rotor blades.

[0009] In this turbine shroud support configuration of the present invention, the means for supporting the respective shroud segments in place comprises a plurality of shroud support segments forming an annular ring assembly to secure the turbine shroud assembly within the shroud housing. The annular ring assembly defines a second cooling air passage for directing a second cooling air flow to cool the annular ring assembly and the turbine shroud segments assembly.

[0010] In accordance with another aspect of the present invention, there is provided a turbine shroud cooling system used in a gas turbine engine having a turbine shroud assembly including a plurality of turbine shroud segments radially spaced apart from a plurality of turbine rotor blades, for controlling tip clearance therebetween. The turbine shroud cooling system comprises a first cooling air passage for selectively directing a first cooling air flow to cool a turbine shroud support assembly, thereby controlling the tip clearance between the turbine shroud segments and the turbine rotor blades. The first cooling air passage is isolated from a combustion gas path defined within the turbine shroud assembly and is in fluid communication with a downstream cooling air passage of the gas turbine engine for directing the first cooling air flow to cool a turbine component downstream of the turbine rotor blades after the first cooling air flow cools the turbine shroud support assembly.

[0011] The turbine shroud cooling system of the present invention preferably comprises a second cooling air passage in fluid communication with a combustion gas path defined within the turbine shroud assembly, for directing a second cooling air flow to cool the turbine shroud assembly and further discharging the second cooling air flow into the

combustion gas path after the second cooling air flow cools the turbine shroud assembly.

[0012] In accordance with a further aspect of the present invention, there is provided a method used with a gas turbine engine for controlling tip clearance between a plurality of turbine rotor blades and a turbine shroud assembly including a plurality of turbine shroud segments radially spaced apart from the respective turbine rotor blades. The method of the present invention comprises directing a first cooling air flow to cool a turbine shroud support assembly, thereby controlling the tip clearance between the turbine shroud segments and the turbine rotor blades and then directing the entire amount of the first cooling air flow that has cooled the turbine shroud support assembly, further to cool the turbine component downstream of the turbine rotor blades.

[0013] The method of the present invention preferably comprises a step of directing a second cooling air flow independent from the first cooling air flow, to cool the turbine shroud assembly, and further discharging the second cooling air flow into a combustion gas path defined within the turbine shroud assembly.

[0014] In one embodiment of the present invention, the second cooling air flow is introduced from a full pressure compressor air flow and cools the turbine shroud assembly continuously during the engine operation including a startup period thereof so that the turbine shroud segments will generally thermally respond to the full pressure compressor air temperature. Then, when the gas turbine engine has thoroughly stabilized and the pinch-point has been avoided, the first cooling air flow is used to cool the annular shroud housing. This first cooling air flow is introduced from the inter-stage compressor bleed air and can be regulated, for example by a control valve. Using inter-stage compressor

bleed air for cooling the annular shroud housing provides more flexibility for tuning turbine tip clearance because the inter-stage compressor bleed air is from a lower temperature cooling source. The first cooling air flow, after being used to cool the annular shroud housing, is not wasted but is substantially re-used for cooling, for example the low pressure turbine stage one vanes. The benefit of re-using the first cooling air flow lies in that it minimizes parasitic secondary air system losses. It should also be noted that the double shroud support assembly configuration is adapted to accommodate the large thermal gradient from the gas path to the turbine support case, which contributes to achieving lower turbine support assembly temperatures, in contrast to prior art passive cooling systems. Therefore, overall engine performance is improved.

[0015] Other advantages and features of the present invention will be better understood with reference to a preferred embodiment described hereinafter.

BRIEF DESCRIPTION OF THE DRAWINGS

[0016] Having thus generally described the nature of the present invention, reference will now be made to the accompanying drawings, showing by way of illustration the preferred embodiment thereof, in which:

[0017] Fig. 1 is a longitudinal cross-sectional schematic view of a gas turbine engine incorporating one embodiment of the present invention;

[0018] Fig. 2 is a longitudinal cross-sectional of a turbine shroud support configuration used in the embodiment shown in Fig. 1; and

[0019] Fig. 3 is an enlarged center portion of Fig. 2, more clearly illustrating the features of the invention.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENT

[0020] Referring to the drawings, particularly Fig. 1, a exemplary gas turbine engine 10 includes in serial flow communication about a longitudinal central axis 12, a fan having a plurality of circumferentially spaced apart fan or rotor blades 14, a conventional low pressure compressor 16, a conventional high pressure compressor 18, a conventional annular combustor 20, a high pressure turbine 22 which includes a turbine shroud support configuration 100 according to one embodiment of the present invention, and a conventional low pressure turbine 24. The low pressure turbine 24 is securely connected to both the low pressure compressor 16 and the fan blades 14 by a first rotor shaft 26, and the high pressure turbine 22 is securely connected to the high pressure compressor 18 by a second rotor shaft 28. Conventional fuel injecting means 30 are provided for selectively injecting fuel into the combustor 20, for powering the engine 10.

[0021] A conventional annular casing 32 surrounds the engine 10 from the low pressure compressor 16 to the low pressure turbine 24, and defines, with the low pressure compressor 16, a low pressure compressor inlet 34 for receiving a portion of ambient air 36. The downstream end of the casing 32 defines with a conventional annular exhaust plug 40, an annular exhaust outlet 42. A portion of the air 36 compressed by the fan blades 14 adjacent to the blade roots 38, is further compressed by the low pressure compressor 16 and the high pressure compressor 18, to be forced into the combustor 20. The mixture of the compressed air 36 and the fuel injected by the fuel injecting means 30, generates combustion gases 52. The combustion gases 52 cause the high pressure turbine 22 and the low pressure turbine 24 to rotate respectively, for powering the high pressure compressor 18, the low pressure compressor 16, and the fan blades 14. Surrounding the fan

blades 14 and the upstream portion of the casing 32, is a short cowl nacelle 44 which is spaced radially outwardly from the casing 32 to define with the casing 32, an annular duct 55 for permitting the radially outer portion of the air 36 compressed by the fan blades 14 to bypass the engine 10. A plurality of circumferentially spaced stator vanes 46 extend radially between the casing 32 and the nacelle 44, and are axially spaced apart downstream of the fan blades 14. The nacelle 44 includes an inlet 48 at its upstream end for receiving the ambient air 36, and an outlet 50 for discharging the portion of air 36 which has been compressed by the fan blades 14 and passed over the stator vanes 46, in order to provide a portion of thrust.

[0022] Inter-stage compressor bleed air is introduced, for example by an air passage which is schematically shown and indicated by numeral 58, to the turbine shroud support configuration 100 for cooling same and thereby controlling turbine tip clearance. A switching valve 60 is provided for controlling the inter-stage bleed air passing through the passage 58. Full pressure compressed air is also introduced, for example by a passage which is schematically shown and indicated by numeral 62, to the turbine shroud support configuration 100.

[0023] The turbine shroud support configuration 100 according to one embodiment of the present invention as illustrated in Fig. 2 includes an integral annular shroud housing 102 secured within a turbine support casing 104 which is part of the annular casing 32 shown in Fig. 1. A number of the components of the turbine shroud support configuration are more clearly illustrated in an enlarged scale, as shown in Fig. 3. The annular shroud housing 102 generally includes an upstream axial section 106 and a downstream axial section 108. The upstream axial section 106 has an external diameter

slightly smaller than the inner diameter of the turbine support casing 104 and includes an external radial flange 110. Thus, the upstream axial section 106 of the annular shroud housing 102 can be appropriately supported within the shroud support casing 104 when the external radial flange 110 thereof is sandwiched between flanges 112 of two sections of the turbine support casing 104 and is secured by mounting screws 114.

[0024] The downstream axial section 108 of the annular shroud housing 102 has a diameter smaller than the diameter of the upstream axial section 106 and is connected to the upstream axial section 106 by a radial front wall 116 so that the downstream axial section 108 is radially spaced apart from the turbine support casing 104. An aft radial wall 118 is provided to the downstream axial section 108 at an aft end thereof, and abuts an inner radial wall 120 of the turbine support casing 104, thereby forming an annulus 122 defined between the turbine support casing 104 and the downstream axial section 108 of the annular shroud housing 102. The annulus 122 is axially aligned with a plurality of air passages 124 passing through a number of support vanes 126 which are circumferentially spaced apart from one another to support the turbine support casing 104 within the engine structure. The air passages 124 are in fluid communication with air passage 58 shown in Fig. 1, for introducing the inter-stage compressor bleed air, as indicated at numeral 174. A plurality of openings 128 are defined in the aft radial wall 118 of the annular shroud housing 102, in fluid communication with a downstream air passage 130 which is adapted to direct a cooling air flow to a downstream turbine component, for example a plurality of low pressure turbine stage one vanes 132.

[0025] An annular impingement skin 134 is provided within the annulus 122 and is secured at opposed ends to the respective front and aft radial walls 116 and 118. The impingement skin 134 includes a plurality of small holes therein (not shown) to permit cooling air 174 under pressure, to pass therethrough, thereby forming fine air jets impinging on the downstream axial section 108 of the annular shroud housing 102. A plurality of fins 136 are provided on the external surface of the downstream axial section 108 of the annular shroud housing 102, to increase contact areas with the cooling air flow and thereby improve its cooling efficiency.

[0026] The front radial wall 116 of the annular shroud housing 102 includes securing devices 138 for secure connection to a plurality of high pressure turbine vanes 140. An annular front hook 142 and an annular rear hook 144 are provided to the inner surface of the downstream axial section 108 of the annular shroud housing 102, for supporting a plurality of shroud support segments 146. The shroud support segments 146 form an annular ring assembly to secure the turbine shroud assembly which is formed by shroud segments 148 within the annular shroud housing 102. Each shroud support segment 146 includes front and aft radial walls 150 and 152, interconnected by an axial wall 154. Hooks 156 and 158 are provided at the top of respective front and aft radial walls 150 and 152, and engage the respective hooks 142 and 144 of the annular shroud housing 102, so that each shroud support segment 146 is securely supported within the annular shroud housing 102. A front leg 160 extending from a lower end of the front radial wall 150 engages a hook 162 of a corresponding shroud segment 148, and an aft leg 164 is securely attached to an aft leg 166 of the corresponding shroud segment 148 by a conventional C-clip 168. Thus, the shroud segments 148 are securely installed within an annular

ring assembly formed by the shroud support segments 146 which in turn are securely supported within the annular shroud housing 102. The axial position of the shroud support segments 146 are restrained between the rear hook 144 of the annular shroud housing 102 and a support structure attached to the high pressure turbine vanes 140.

[0027] Openings (not indicated) provided in each shroud support segment 146, together with clearances (not shown) between adjacent shroud support segments 146, form an air passage which is in fluid communication with combustion gas path 170 defined within the turbine shroud assembly, and is also in fluid communication with the air passage 62 shown in Fig. 1, so that full pressure compressor-delivered air, as indicated at numeral 172, can flow through the shroud support segments 146 to cool both the shroud support segments 146 and the shroud segments 148, independently from the cooling air flow 174 which is isolated from joining the air flow 172 by the annular shroud housing 102, and is then discharged through the air passages in and between the shroud segments 148, into the combustion gas path 170.

[0028] Referring now to both Figs. 1 and 2, during operation, full pressure compressor-delivered air 172 is introduced through the air passage within the shroud support segments 146, to cool both the shroud support segments 146 and the shroud segments 148 during engine startup, and is then continuously supplied during the entire engine operation process. The air flow 172 also functions as an air seal around the shroud segments 148 and is then forced to pass through the passages in and between the shroud segments 148, surrounding the turbine blade tips before being discharged through the combustion gas path 170, in order to prevent combustion gas leakage. Therefore, the air flow 172 requires relatively high pressure and the full pressure

compressor-delivered air is a preferable source, although the temperature thereof is relatively high, which reduces its cooling efficiency and thereby its flexibility for turbine tip clearance control. Nevertheless, this problem is addressed with the use of cooling air flow 174.

[0029] The switching valve 60 is connected in the air passage 58 and has an "on" position and preferably an "off" position. When the switching valve 60 is in the "on" position, the air flow passing through the air passage 58 is diverted into branch air passage 58a with preferably 50 percent flow thereof, and into branch air passage 58b with preferably the other 50 percent flow thereof. When the switching valve 60 is in the "off" position, the branch air passage 58a is shut off and the entire air flow from air passage 58 is directed into branch air passage 58b. Although a complete 'shut-off' of passage 58a is preferred here, for reasons described below, it is not necessary and the respective flows through passages 58a and 58b can be selected by the designer as desired.

[0030] Generally, during the engine startup period ("startup period" being used to refer here to an engine start, run-up or other transient operating condition in the engine operating cycle), the switching valve 60 is in the "off" position and the inter-stage compressor bleed air from passage 58 passes through the branch air passage 58b to cool the downstream components of the turbine, such as low pressure turbine (LPT) stator and/or vanes 132. When the engine has reached cruise and stabilized thermally such that the pinch-point has been avoided, the switching valve 60 is activated to its "on" position so that about 50 percent (preferably) of the inter-stage compressor bleed air flow is directed from passage 58 through branch air passage 58a, to provide the cooling air flow 174 for cooling the annular

shroud housing 102. Meanwhile, the remaining inter-stage compressor bleed air flow is directed through branch air passage 58b to continue cooling the downstream turbine components. The cooling air flow 174 is directed to pass through the impingement skin 134 (as represented by arrows 175 in Figs. 2 and 3) to thereby impinge on the annular shroud housing 102 and then flow along the external surface of the annular shroud housing 102 (as represented by arrow 176 in Figs. 2 and 3), passing fins 136 thereof to further cool the annular shroud housing 102, before being discharged through the downstream air passage 130 (as represented by arrow 177 in Figure 2) in order to cool the downstream turbine components such as the low pressure turbine [LPT] vanes and/or stator 132. The air flow 174 is not discharged into the combustion gas path 170 and therefore requires only a relatively low pressure (relative to the P3 flow) to deliver the air flow 174 for cooling the engine components until the pressure is completely lost. The inter-stage compressor bleed air has a relatively lower temperature and a low air pressure, and is therefore a preferable source of cooling air 174 than using P3 air, when possible. Thus, the cooling air flow 174 not only provides an additional cooling, with respect to the cooling provided by air flow 172, to the entire turbine shroud and support structure to improve cooling efficiency, but also provides more flexibility for tuning turbine tip clearance because of the relatively low temperature of the cooling air source. The re-use of the shroud cooling air flow 174 advantageously minimizes parasitic secondary air system loses of engine performance.

[0031] The switching valve 60 can be any suitable valving or switching, or other means for controlling the flow of air directed to provide turbine tip clearance cooling as described above. The switching valve 60 can be controlled at

any time during the engine operation, to control the turbine tip clearance during various engine operative conditions.

[0032] Passages 58 and 62 in Fig. 1 are exemplary, schematically illustrating the respective cooling air sources, and are not intended to be limited to any particular structural arrangement for obtaining the respective inter-stage compressor bleed air (i.e. P2.X) and full pressure (i.e. P3) compressor delivered air. It will be understood that these can be achieved using a variety of known arrangements.

[0033] One skilled in the art will understand, in light of this disclosure, that switching valve 60 may be replaced by any functional equivalent which permits the air flow through air passage 58a to be controlled, restricted or stopped, as desired by the designer. For example, a simple open/closed valve or other flow control member may be placed downstream of the branch between passages 58a and 58b. Other configurations will also be apparent to the skilled reader and thus are not intended to be outside the scope of the present disclosure.

[0034] The cooling system and turbine tip clearance control method of the present invention is not applied only to the short cowl nacelle engines which are taken as an example to illustrate the applications of the present invention. The present invention can be applied to various types of gas turbine engines without departing from the spirit of this invention. Though the use of P2.X interstage compressor air is of course preferred, it is only preferred and thus not necessary.

[0035] Modifications and improvements to the above-described embodiment of the present invention may become apparent to those skilled in the art. The foregoing description is

intended to be exemplary rather than limiting. The scope of the invention is therefore intended to be limited solely by the scope of the appended claims.

I/WE CLAIM:

1. A gas turbine engine having a plurality of turbine rotor blades and a turbine shroud assembly, the turbine shroud assembly including a plurality of turbine shroud segments radially spaced apart from the plurality of turbine rotor blades, the gas turbine engine further comprising:

   an annular shroud housing adapted to be secured within a turbine support casing;

   turbine shroud segment attachment members mounted to an inner side of the annular shroud housing and adapted to support the turbine shroud segments in place;

   a first cooling air passage adapted to direct a first cooling air flow from a compressor portion of the gas turbine engine to at least one gas turbine engine component downstream of the shroud housing relative to a combustion gas path through the gas turbine engine;

   a second cooling air passage branching from the first cooling air passage and adapted to direct a second cooling air flow from the first cooling passage to the shroud housing to cool the shroud housing and thereby affect the tip clearance between the turbine shroud segments and the turbine rotor blades; and

   a flow control member associated with the second cooling air passage and adapted to selectively control a cooling air flow passing through the second cooling air passage, the flow control member cooling air being selectively positionable between a first position, in which a first cooling air flow

rate is permitted to pass through the second cooling air passage, and a second position, in which a second cooling air flow rate is permitted to pass through the second cooling air passage.

2. A gas turbine engine as claimed in claim 1 wherein, the first cooling air passage is defined at least by the shroud housing, and is isolated from a section of the combustion gas path defined within the turbine shroud assembly

3. A gas turbine engine as claimed in Claim 2 wherein the second cooling air passage is in fluid communication with a downstream cooling air passage of the gas turbine engine so that the cooling air flow passing to the shroud housing from the second cooling air passage is redirected therefrom to further cool a turbine component downstream of the turbine rotor blades relative to the combustion gas path.

4. A gas turbine engine as claimed in Claim 3 wherein the second cooling air flow rate is substantially zero.

5. A gas turbine engine as claimed in claim 1 wherein the turbine shroud segment attachment members comprise a plurality of shroud support segments forming an annular ring assembly to secure the turbine shroud assembly within the shroud housing, the annular ring assembly defining a third cooling air passage for directing a third cooling air flow to cool the annular ring assembly and the turbine shroud segments.

6. A gas turbine engine as claimed in claim 5 wherein the third cooling air passage is adapted to be in fluid communication with the combustion gas path defined

within the turbine shroud assembly so that the third cooling air flow is adapted to be discharged into the combustion gas path after having cooled the turbine shroud assembly.

7. A gas turbine engine as claimed in claim 6 wherein the first cooling air passage is adapted to be in fluid communication with an upstream cooling air passage for intake of a compressor bleed air flow, and wherein the third cooling air passage is adapted to be in fluid communication with an upstream cooling air passage for intake of full pressure compressor air to form the third cooling air flow.

8. A turbine shroud cooling system used in a gas turbine engine having a turbine shroud assembly including a plurality of turbine shroud segments radially spaced apart from a plurality of turbine rotor blades, for controlling tip clearance therebetween, the turbine shroud cooling system comprising:

a second cooling air passage for selectively directing a first cooling air flow to cool a turbine shroud support assembly, thereby controlling the tip clearance between the turbine shroud segments and the turbine rotor blades; and

the second cooling air passage being isolated from a combustion gas path defined within the turbine shroud assembly, and being in fluid communication with a downstream cooling air passage of the gas turbine engine for directing the second cooling air flow to cool a turbine component downstream of the turbine rotor blades after the second cooling air flow cools the turbine shroud support assembly.

9. A turbine shroud cooling system as claimed in claim 8 wherein the second cooling air passage branches from a first cooling air passage which is adapted to receive a compressor bleed air flow to form a first cooling air flow to cool a turbine component downstream of the turbine rotor blades.

10. A turbine shroud cooling system as claimed in claim 9 further comprising controlling means for selectively directing a proportion of the entire cooling air flow in the first cooling air passage to the second cooling air passage.

11. A Turbine shroud cooling system as claimed in claim 10 wherein the controlling means are adapted to direct a proportion between zero and a predetermined percentage of the entire cooling air flow in the first cooling air passage, thereby maintaining the downstream turbine component to be cooled.

12. A turbine shroud cooling system as claimed in claim 8 further comprising a third cooling air passage in fluid communication with the combustion gas path defined within the turbine shroud assembly, for directing a third cooling air flow to cool the turbine shroud assembly and discharging the third cooling air flow into the combustion gas path after the third cooling air flow cools the turbine shroud assembly.

13. A turbine shroud cooling system as claimed in claim 12 wherein the third cooling air passage is adapted for receiving a full pressure compressor air flow to form the third cooling air flow.

14. A method for controlling tip clearance between a plurality of turbine rotor blades and a turbine shroud assembly in a gas turbine engine, the shroud assembly including a plurality of turbine shroud segments radially spaced apart from the respective turbine rotor blades, the gas turbine engine having a cooling system adapted to direct a first cooling air flow from a compressor portion of the gas turbine engine to at least one component to be cooled by the cooling air, the method comprising the steps of:

selectively diverting a second cooling air flow from the first cooling air flow; and then

directing the second cooling air flow to cool a turbine shroud support assembly to thereby affect the tip clearance between the turbine shroud segments and the turbine rotor blades,

wherein the second cooling air flow is selectively diverted depending on an operating condition of the gas turbine engine.

15. A method as claimed in claim 14 further comprising the step of directing the second cooling air flow to further cool a turbine component downstream of the turbine rotor blades.

16. A method as claimed in claim 14 wherein the operating condition corresponds to engine run-up.

17. A method as claimed in claim 14 comprising a step of directing a third cooling air flow independent from the first and second cooling air flows, to cool the turbine shroud assembly, and further discharging the third cooling air flow into a combustion gas path defined within the turbine shroud assembly.

18. A method as claimed in claim 17 comprising a step of receiving a full pressure compressor air flow to form the third cooling air flow.

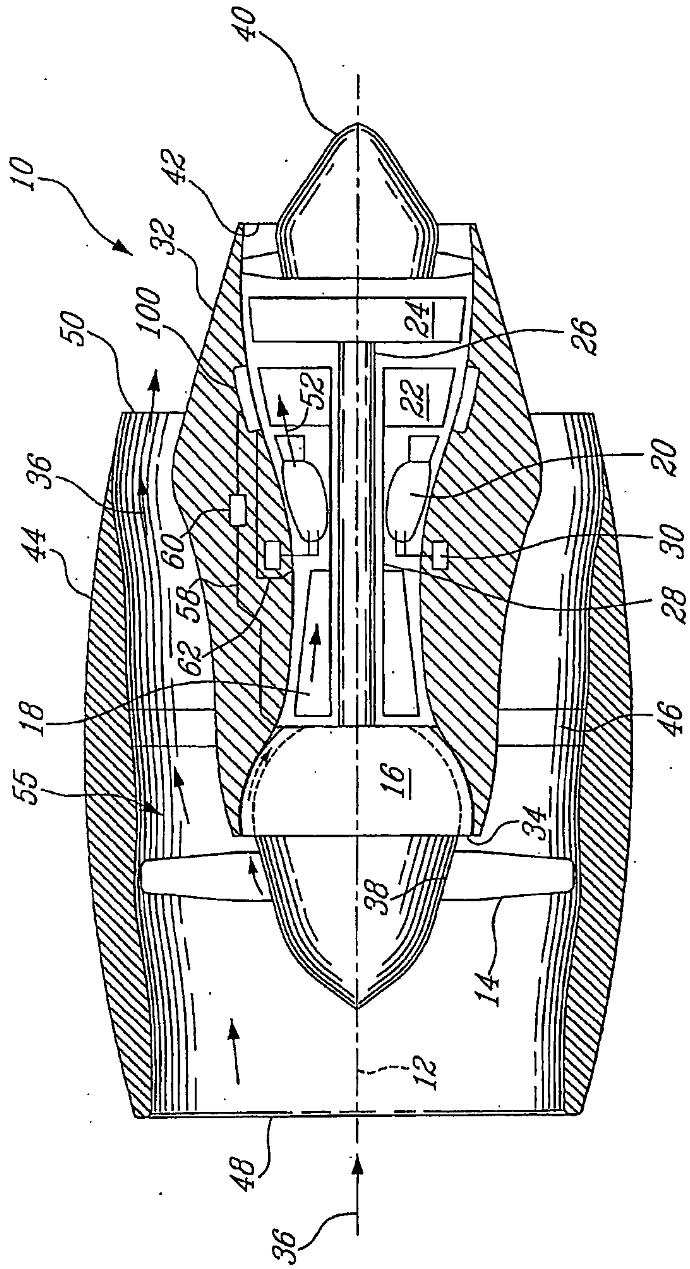
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Fig-1

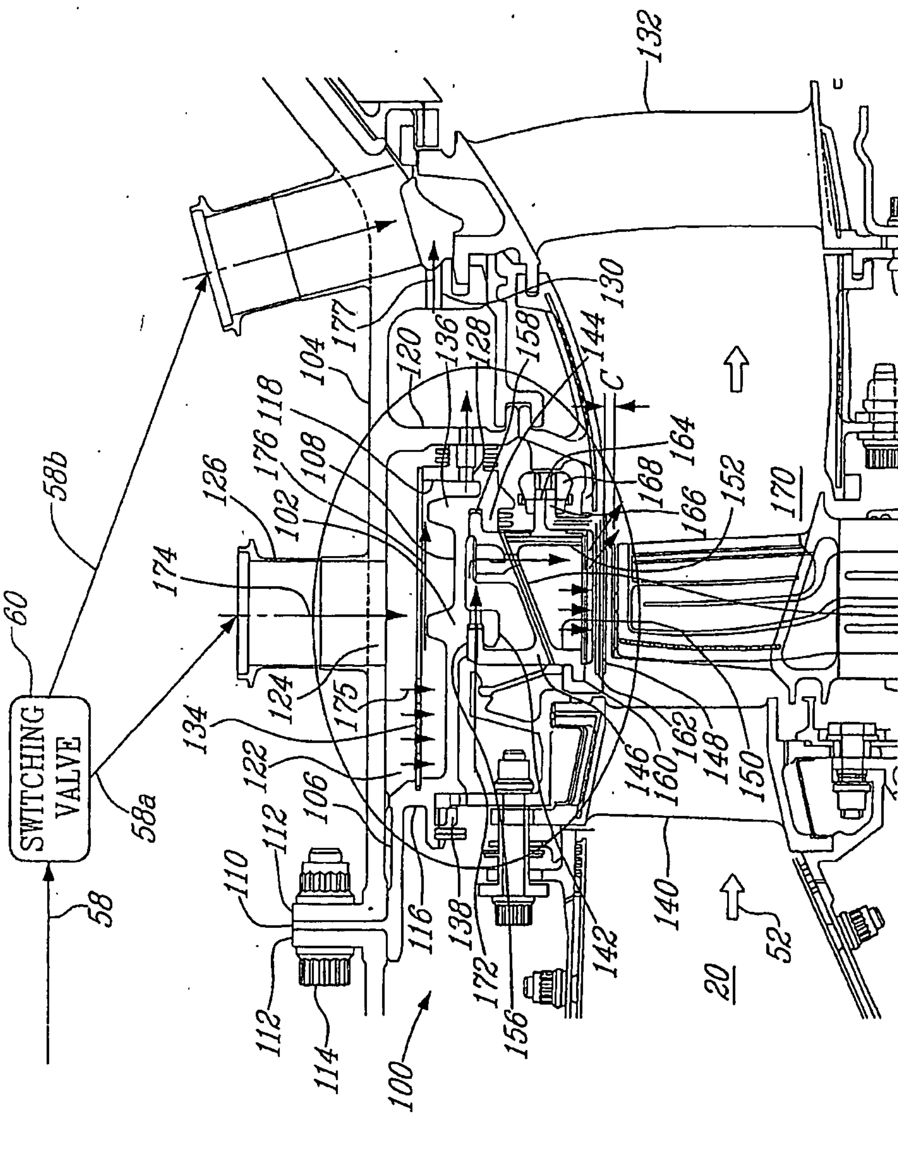
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FIG-2

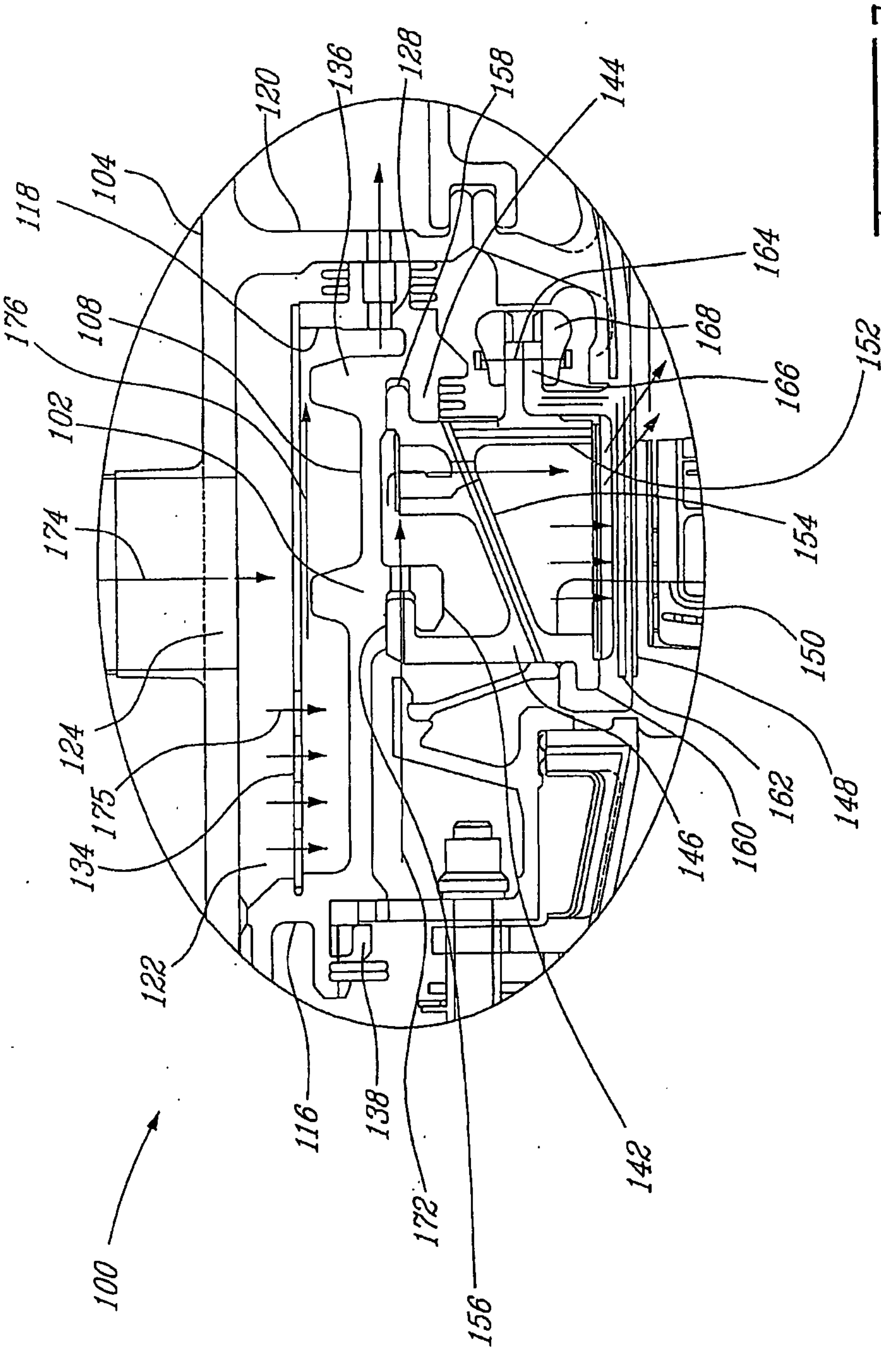
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FIG. 3

INTERNATIONAL SEARCH REPORT

International Application No
PCT/CA2004/000563

A. CLASSIFICATION OF SUBJECT MATTER
IPC 7 F01D11/24 F01D11/10

According to International Patent Classification (IPC) or to both national classification and IPC

B. FIELDS SEARCHED

Minimum documentation searched (classification system followed by classification symbols)
IPC 7 F01D

Documentation searched other than minimum documentation to the extent that such documents are included in the fields searched

Electronic data base consulted during the international search (name of data base and, where practical, search terms used)

EPO-Internal

C. DOCUMENTS CONSIDERED TO BE RELEVANT

Category °	Citation of document, with indication, where appropriate, of the relevant passages	Relevant to claim No.
X	GB 1 581 855 A (GEN ELECTRIC) 31 December 1980 (1980-12-31)	1-5, 8-11, 14-16
Y	page 1, line 17 - line 38 page 1, line 79 - line 91 page 2, line 26 - page 3, line 13 page 3, line 87 - line 94 figure 1 -----	6,7,12, 13,17,18
Y	US 5 993 150 A (LIOTTA GARY C ET AL) 30 November 1999 (1999-11-30) column 3, line 27 - column 5, line 30; figures 1,2 -----	6,7,12, 13,17,18

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[X] Further documents are listed in the continuation of box C.

[X] Patent family members are listed in annex.

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"X" document of particular relevance; the claimed invention cannot be considered novel or cannot be considered to involve an inventive step when the document is taken alone

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Date of the actual completion of the international search	Date of mailing of the international search report
19 August 2004	30/08/2004

Name and mailing address of the ISA	Authorized officer
European Patent Office, P.B. 5818 Patentlaan 2 NL – 2280 HV Rijswijk Tel. (+31-70) 340-2040, Tx. 31 651 epo nl, Fax: (+31-70) 340-3016	de Rooij, M

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INTERNATIONAL SEARCH REPORT

International Application No
PCT/CA2004/000563

C.(Continuation) DOCUMENTS CONSIDERED TO BE RELEVANT

Category °	Citation of document, with indication, where appropriate, of the relevant passages	Relevant to claim No.
X	GB 1 581 566 A (GEN ELECTRIC) 17 December 1980 (1980-12-17) page 2, line 3 - page 3, line 55 page 5, line 10 - page 6, line 5 figures 1,3,5	1-5, 8-11, 14-16
X	US 3 975 901 A (HALLINGER CLAUDE CHRISTIAN ET AL) 24 August 1976 (1976-08-24) figures 4a,4b,4c	1,2,5, 9-11, 14-16
A	US 4 849 895 A (KERVISTIN ROBERT) 18 July 1989 (1989-07-18) column 1, line 67 - column 2, line 44; figures 2,3 column 3, line 7 - column 4, line 4	1-18
A	US 5 048 288 A (BESSETTE ALAN D ET AL) 17 September 1991 (1991-09-17) column 2, line 10 - line 56 column 3, line 50 - column 5, line 8; figures	1-18
A	US 4 679 981 A (GUIBERT ALAIN J E ET AL) 14 July 1987 (1987-07-14) column 4, line 42 - column 5, line 7 column 6, line 44 - line 64; figure 2	1-18
A	US 4 596 116 A (MANDET GERARD M F ET AL) 24 June 1986 (1986-06-24) the whole document	1-18

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INTERNATIONAL SEARCH REPORT

Information on patent family members

International Application No
PCT/CA2004/000563

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